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1. The first step in the process is to identify the problem or issue that needs to be addressed. This involves gathering information and understanding the context of the problem.

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**DEPARTMENT OF DEFENCE
DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION
AERONAUTICAL RESEARCH LABORATORY**

Propulsion Report 187

**GENERAL REQUIREMENTS AND TECHNIQUES
FOR COMPONENT FATIGUE LIFE SUBSTANTIATION
IN AUSTRALIAN SERVICE HELICOPTERS**

by

K.F. FRASER

SUMMARY

An Australian Defence Force requirement has been defined to provide in-country capability to support component fatigue life substantiation in selected Australian fleet helicopters, with initial application to the S-70A-9 Black Hawk helicopter operated by the Australian Army. The implications of this requirement are examined, and the need to assess the severity of the spectrum of normal missions for the selected aircraft fleet is supported. Justification for a program to assess mission severity from measurements of flight regime recognition data and loads in selected components in a sample of fleet aircraft, is provided. A program to substantiate the fatigue lives of selected Black Hawk helicopter components, subject to significant in-service loads, is outlined. The general requirements of airborne and ground data systems required in support of the program are examined.



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TABLE OF CONTENTS

PAGE NOS.

ABBREVIATIONS	iii
1. INTRODUCTION	1
2. COMPARISON WITH FIXED-WING AIRCRAFT	2
3. HELICOPTER LOAD ENVIRONMENT	3
4. HELICOPTER COMPONENT FATIGUE LIFE SUBSTANTIATION	9
4.1 Life Estimation Methodology and Need for Reliable Estimation	9
4.2 Assumed Usage Spectrum	10
4.3 Measured Usage Spectrum for Sample of Fleet Aircraft	11
4.4 Individual Aircraft Usage Monitoring	12
5. COMPONENT FATIGUE LIFE SUBSTANTIATION PROGRAM REVIEW ..	13
5.1 General Comments	13
5.2 UK Military Helicopter Programs	13
5.3 UK Civil Helicopter Programs	15
5.4 USA Military Helicopter Programs	15
6. DATA REQUIRED FOR FATIGUE SUBSTANTIATION	17
6.1 General Comments	17
6.2 Fleet Utilization Data	17
6.3 Configuration Details	17
6.4 Flight Regime Recognition Data	18
6.5 Loads Inferred by Indirect Measurement	18
6.6 Directly Measured Loads	19
7. PARAMETER LIST	19
7.1 General Comments	19
7.2 Flight State Recognition Parameters	20
7.3 Engine Parameters	20
7.4 Powertrain Parameters	21
7.5 Rotor System Parameters	22
7.6 Rotor Pitch Control System Parameters	23
7.7 Airframe Parameters	23
7.8 Other Parameters	23

TABLE OF CONTENTS CNT.

8.	FUNCTIONAL REQUIREMENTS OF DATA SYSTEM	24
8.1	General Requirements	24
8.2	Input Parameter Sensing	24
8.3	Signal Conditioning	24
8.4	Programmable Signal Acquisition	25
8.5	Manual Input	26
8.6	Airborne Data Storage	26
8.7	Flight Line Test	27
8.8	Data Recovery and Verification	27
8.9	Data Collation and Analysis	28
8.9.1	Sikorsky Data Collation Requirement	28
8.9.2	WHL Data Collation Requirement	29
8.9.3	Graphical Presentation and Archiving of Data	30
8.9.4	Data Analysis	30
9.	CHOICE OF DATA SYSTEM	30
10.	CONCLUDING REMARKS	31
	ACKNOWLEDGEMENT	32
	REFERENCES	33-35
	TABLES 1 - 8	
	FIGURES 1 - 3	
	DISTRIBUTION	
	DOCUMENT CONTROL DATA	

ABBREVIATIONS

AC Alternating current
 ADF Australian Defence Force
 ARL Aeronautical Research Laboratory (Melbourne, Australia)
 AUW All up weight
 BIT Built-in-test
 CAA Civil Aviation Authority (UK)
 CM Canadian Marconi
 COG Centre of gravity
 DC Direct current
 EDS Enhanced Diagnostic System (MDHC)
 EEL EEL Limited (a Division of Westland Aerospace)
 EULMS Engine Usage Life Monitoring System (Plessey)
 GAG Ground-air-ground cycle
 GE General Electric
 GSE Ground support equipment
 HCF High cycle fatigue
 HODR Helicopter Operational Data Recording (MOD program)
 HUM Health and usage monitoring
 LCF Low cycle fatigue
 LRU Line replaceable unit
 MDHC McDonnell Douglas Helicopter Company
 MOD Ministry of Defence (UK)
 MODAS Modular Data Acquisition System (Plessey)
 MRGB Main rotor gear box
 MSA Mission severity assessment
 PC Personal computer
 RAAF Royal Australian Air Force
 RAE Royal Aircraft Establishment (UK)
 RAF Royal Air Force
 RAM Random access memory
 RAN Royal Australian Navy
 RN Royal Navy
 ROM Read only memory
 RPM Revolutions per minute
 RTM Rolls-Turbo-Meca
 SUM Structural Usage Monitor (Canadian Marconi)
 SUMS Structural Usage Monitoring System (Plessey)
 US United States
 WHL Westland Helicopters Limited

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1. INTRODUCTION

Maintaining helicopter component integrity is a major requirement for operators. The main problem area is within the highly stressed dynamic system, comprising rotor systems and controls, powertrain (transmission system), and engines. Much of the dynamic system cannot be duplicated with presently available technology.

Some components subject to high amplitude cyclic stresses may eventually undergo fatigue failure if a sufficiently high number of load cycles is applied before the component is replaced. For metallic load bearing components, which are commonplace in helicopters, fatigue strength is the dominant design parameter¹. The nature of the helicopter load environment is such that, for helicopters currently in service, there are normally some components, particularly within the dynamic system, for which the risk of such failure would increase to an unacceptable level if the components were left in service indefinitely. Such components must be replaced before the risk of fatigue failure exceeds an acceptable limit. Fatigue failure of dynamic system components has been a major cause of airworthiness-related accidents in helicopters² and much emphasis is placed by the helicopter community on minimizing the risk of in-flight component fatigue failures.

Achieving a high level of integrity assurance in aircraft involves very high maintenance and repair costs. There is scope for considerable savings in the cost of helicopter maintenance with improved lifing methods which would allow many components to be operated for longer periods within assured safety margins.

Generally helicopter component replacement is undertaken according to either a health monitoring indication or after a designated safe life has expired. Health monitoring involves the detection of a changed or abnormal condition, such as may be observed in respect of wear particles in lubricating oil or a change in a vibration signal. In-service lives of components can be optimized if reliable health monitoring techniques, which provide adequate warning of impending failures, can be implemented. In cases where components are prone to fracture by fatigue failure and the life after crack initiation is too low to permit reliance on health monitoring techniques, or available health techniques are inadequate, some plan for replacing components when their safe lives have expired is essential. Replacement strategies for components in this latter category will be considered in this report.

It is common practice to specify the fatigue life of a helicopter component as the maximum number of in-flight hours permissible before the component is replaced (ie. the "safe life" approach). The substantiation of the defined safe lives of fatigue life limited components has been considered very desirable by many operators and a number of programs, involving in-flight measurements, have been undertaken to deal with this issue. The justification for an Australian program to support helicopter component fatigue life substantiation is addressed in this report.

Australian military helicopters currently in service or to come into service in the near future comprise Black Hawk, Seahawk, Squirrel, Iroquois, Sea King, Kiowa and Chinook. Except for the Chinook, all helicopters currently in military service in Australia are of the "single main lift rotor" configuration³ (ie. employ a smaller tail rotor in the

vertical plane to provide reaction torque). Consideration will be confined to that type of helicopter (sometimes referred to as the "penny farthing" type) in this report. O'Shea⁴ identifies the helicopters, both military and civil, operated in Australia.

The RAAF and the RAN have defined a joint-service requirement for ARL to establish a local capability to collate appropriate operational data for use in component fatigue life substantiation in service helicopters, with initial consideration being given to the application to the S-70A-9 Black Hawk helicopter. Although this helicopter is operated by the Australian Army, the RAAF has the responsibility for its airworthiness. The S-70B-2 Seahawk operated by the RAN has considerable commonality with the Black Hawk.

General requirements for component fatigue life substantiation in Australian service helicopters are examined in this report. Based on those requirements, a definition of the capabilities demanded of airborne and ground data systems to support a suitable program, is given.

2. COMPARISON WITH FIXED-WING AIRCRAFT

The fixed-wing aircraft is supported by its wing and airframe structure whereas the helicopter is supported by the forces conveyed by a mechanism (the moving components of the rotor and transmission systems). It is generally not feasible to provide many of these components with the same level of integrity assurance which has long been provided in fixed-wing aircraft.

The Working Group on Helicopter Health Monitoring convened by the CAA states⁵ that "the helicopter has been recognized as being fundamentally different to its fixed wing counterpart, evident by the fact that it is implicit in fixed wing requirements that all critical failures shall be sufficiently infrequent, by virtue of integrity or duplication, for operational constraints to be unnecessary". That Group went on to state that "there is less opportunity to duplicate vital transmission and power plant paths to match the level of fixed wing safety" but that "a forced landing can be a less hazardous event than for fixed wing, and will be reasonably survivable provided that the rotorcraft is not over hostile surface". Improved survivability in the event of total loss of engine power arises because of the capacity of rotary-wing aircraft to autorotate to ground level, provided the engine failure occurs at sufficient altitude and provided the rotor/transmission system is intact. Fail-safe or damage tolerant design methods have been commonplace for fixed-wing aircraft for many years. Implementation of damage tolerant design methods for critical helicopter components is far more difficult due to the small size, complexity and high frequency loading of such components. The high frequency loading can give rise to a rapid rate of crack growth. The CAA⁶ strongly encouraged the introduction of damage tolerant design methods for helicopter sub-systems where practical.

Whereas loads may tend to be relatively static during steady flight of a fixed-wing aircraft, those for a helicopter tend to be dynamic and accompanied by high levels of vibration. These matters are considered in some detail by De Jonge⁷.

For metallic load bearing components, which are commonplace in helicopters, fatigue strength is usually the dominant design parameter. Composite materials have been used less in helicopter dynamic systems than in fixed-wing airframes. However their use in rotor blades and hinges has now become widespread, and their use is likely to be extended to other components in the dynamic system and in the airframes in future helicopters. Composite materials generally have a much improved fatigue strength relative to their metallic counterparts. However adequate fatigue strength for composite materials does not imply an adequate static strength margin, and hence more consideration needs to be given to the static strength of composite components. For the S-70A-9 Black Hawk and S-70B-2 Seahawk helicopters, the main rotor blades are composed of a high proportion of composite (non-metallic) material but some metallic elements (spars etc.) are also included; the tail rotor blades are constructed entirely of composite materials and the main rotor hinge bearings are of elastomeric material.

It is normal for load-bearing metallic helicopter components to be designed according to safe-life principles which are discussed in more detail by Krasnowski et al⁸ and Spiegel⁹. Major loads in helicopter transmission and rotor systems are carried by single load path components. This is to be compared with fixed-wing aircraft airframes which are largely designed with redundancy, so that a damage tolerant approach can be taken. If a fatigue crack does occur with the damage tolerant design approach, it can be contained by ground inspection and rectification⁶. With the safe-life approach the component is taken out of service before a designated safe life is exceeded.

For helicopters there is more commonality between military and civil aircraft types, than for fixed-wing aircraft. Many civil helicopters are direct derivatives of military types. Probably the most comprehensive review of helicopter airworthiness and strategic planning for improved airworthiness was conducted in the UK by the CAA⁶ in respect of public transport helicopters.

Accident statistics^{6,10} indicate that airworthiness related accidents are more frequent in helicopters than in fixed-wing aircraft and that fatigue fracture is a major cause of such accidents². These statistics relate mainly to civil registered aircraft but there seems no reason to suggest that the same trend does not apply to military aircraft.

3. HELICOPTER LOAD ENVIRONMENT

The load environment for the single main lift rotor type helicopter, in military service in Australia, will be considered here. Major sub-systems in which critical loads may occur are:

- (a) Engines
- (b) Powertrain (gearboxes, shafts, bearings etc. between engines and rotor system)
- (c) Rotor system (rotor hubs and blades).
- (d) Rotor pitch control system
- (e) Airframe

A typical interconnection of the helicopter rotating machinery is given in Fig. 1a which depicts the arrangement for the Sea King. In general, the helicopter is designed to operate with constant rotor speed irrespective of load conditions, but some variations will occur under transient load conditions. Typically the rotational speed of the main rotor is in the range 3 to 6 Hz (180 to 360 RPM) and that of the tail rotor in the range 15 to 30 Hz (900 to 1800 RPM). Rotational speed stability is achieved through regulation of engine power turbine shaft speed which is stepped down by gearboxes to provide the required rotor speeds. Typical power turbine shaft speed is about 330 Hz (20000 RPM). The powertrain normally includes at least three gearboxes, one transmitting torque to the main rotor and two carrying the torque to the tail rotor. An epicyclic arrangement (Fig. 1b) is favoured for the main rotor gearbox (MRGB) as it is capable of providing a large speed reduction together with a high value of output torque. For twin-engine operation an appropriate means of summing the torques from each engine is required together with freewheel capability to allow operation from a single engine. The tail driveshaft is typically connected at some intermediate point in the MRGB. The intermediate gearbox provides an angular take-off to the tail rotor gearbox and normally has an input to output shaft speed ratio approaching unity. The tail gearbox drives the tail rotor and typically provides a speed reduction ratio of about 2.5.

The load environment for each of the helicopter sub-systems will now be examined briefly:

(a) Engines

Stresses in major engine components are caused by centrifugal loads, gas pressure loads, component fixing loads, torques and differential expansion due to thermal gradients. Although centrifugal stress usually dominates, the other sources cannot be ignored. Following a change in collective pitch lever position (which must be accompanied by an automatic or manual throttle movement and a change in gas generator speed) the centrifugal and most other mechanical loads, and the consequent stresses, rapidly reach stable values appropriate to the new throttle setting. The superimposed thermal stresses however may take some time to stabilize at the new values. For example, following a throttle opening, the temperature at the rim of a turbine disc, which is situated in close proximity to the hot gas stream, increases and rapidly stabilises. At the bore of the same disc, the temperature will also increase but at a slower rate. This leads to variation in thermal gradients and corresponding thermal stresses, so that the minimum or maximum total stress may not be reached until some time after the change in collective pitch. If such changes follow in quick succession, stable conditions may never be reached.

The stress excursions which arise due to excursions in gas generator rotational speed (following collective pitch changes) usually limit the safe lives of the relevant engine rotors (compressors and turbines) because of LCF life usage. Engine manufacturers usually produce algorithms which relate LCF life usage to gas generator rotational speed excursions.

Creep and thermal fatigue are related to temperature/time history and are particularly important for turbine blades. Once again engine manufacturers frequently produce algorithms which effectively relate life usage from these sources to the temperature/time history.

The engine controller maintains the speed of the power turbine fairly constant as the load varies (a requirement to maintain rotor speeds approximately constant) but torque fluctuations will occur in the drive shafts as power levels are varied. The torque fluctuations can give rise to fatigue life usage in shafts. Some form of engine torque sensing and cockpit display is always provided in helicopters.

(b) Powertrain

The various gears and shafts within the powertrain transmit engine power to the main and tail rotors. For shafts inboard of the rotor masts, and for gears, torque loads are of primary significance. For a given power throughput, torque in individual components in the main and tail rotor load paths will of course be inversely proportional to their rotational speed and, for convenience, is usually expressed as a per cent rated value. Torque changes will occur during flight to accommodate the different load conditions which apply for each flight regime (ie. a flying condition defined in fairly simple terms to correspond to various ranges or elements of hover, manoeuvre, ascent, descent and forward flight). The required rotor thrust, and hence torque, will be related to the AUW.

The rotor power start/stop cycle and the torque fluctuations during flight (eg. from hover to manoeuvre and back to hover) will be reflected as corresponding variations in torsional stress in powertrain shafts. These variations need to be considered when estimating fatigue lives of shafts. Torque changes, which accompany changes in collective pitch, will occur in shafts associated with the main rotor. Cyclic torque variations, which occur at rates corresponding to multiples of rotor rotational speed, will be superimposed on these relatively slow changes in torque level. Pavia¹¹ measured the magnitude of the torque fluctuations present in the main rotor mast of a Wessex Mk 31B helicopter and found that the combined variations at main rotor mast frequency and blade passing frequency (rotor rotational speed multiplied by the number of blades) were up to 2% rated torque. Bending stresses in powertrain shafting can also impose fatigue penalties. An example¹² for an early model of the Sea King helicopter is a fatigue limit on a tail rotor driveshaft due to vibratory bending stresses induced through an oil cooler take-off pulley load.

For gears the main stress of interest is that arising at the tooth root due to bending. For a simple gear mesh each tooth will receive one cycle of stress per revolution of the respective gear, and hence cyclic loading will occur even when a steady value of torque is transmitted. The rate of variation in transmitted torque (due to changes in mean value or to the presence of alternating components) will be such that the torque may be considered to be constant over any given load cycle for a gear tooth. Some critical gears may be prone to HCF life usage and therefore have fatigue limited lives.

Weight restrictions on gearboxes do not permit an over-conservative approach to gearbox design. The maximum engine torque available under contingency conditions, particularly in twin-engine helicopters, is frequently greater than the maximum torque which can be passed through the gearbox without incurring fatigue life usage on one or more gearbox components. The maximum permissible torque which a particular gear or shaft can transmit without incurring fatigue life usage is usually referred to as the torque endurance limit. Helicopter gearbox design has advanced to the stage that it is usual for all normal helicopter operations to be performed without any gear fatigue life usage, and for some

high durability gearboxes there is insufficient contingency engine power available to incur any gear fatigue life usage. Helicopter manufacturers usually specify transmission contingency limits which designate overtorque boundaries. The lowest contingency limits usually refer to the magnitude of torque above which fatigue damage to one or more gears is predicted. Maximum operating times above such limits are specified and pilots are required to log times above contingency limits. Sometimes a number of limits are specified with reduced allowable operating time at the higher limits. A sample contingency transmission torque specification for the Mk 50 Sea King is given in Table 1. It is quite feasible to automatically limit the maximum engine torque available but, under genuine emergency conditions, it is better to risk fatigue damage to gears than to risk the loss of an aircraft. The Seahawk (US Navy and RAN versions) is fitted with a contingency switch which allows pilots to engage automatic torque limiting in non-hazardous situations. Pilots would normally disengage the automatic function at times when rapid access to high levels of torque may be required (eg. take-off and landing).

The individual powertrain components for a twin-engine helicopter are subject to one of the following five torque levels (expressed in non-dimensional form as per cent rated torque):

- (1) Engine 1 torque.
- (2) Engine 2 torque.
- (3) Total torque (sum of engine 1 torque and engine 2 torque).
- (4) Tail drive torque.
- (5) Total torque minus tail drive torque.

The gear reduction stages ahead of the summing gear are subject to the respective engine torques. The summing gear is unique in that each tooth experiences two load cycles (one from each engine) per revolution of that gear. Some MRGB gears ahead of the tail rotor take-off will experience the total torque while the main rotor shaft and final reduction stage (usually epicyclic) will experience the total torque minus the tail drive torque. The tail rotor may typically require in the order of 12% of total torque under steady conditions. Gearbox inefficiencies and energy extraction for accessories will modify the powertrain torque distribution by a small amount.

The main rotor mast, which is usually (but not always) integral with the main rotor gearbox, can be subject to bending stresses which, in combination with the longitudinal and torsional stresses, can produce significant overall stress levels at some locations. Aerodynamic forces on the blades can give rise to significant cyclic bending loads in the region of the upper bearing. During forward flight the main rotor blades will be subject to variations in flapping angle throughout a revolution of the main rotor. The alternating bending stress is related to the changes in flapping angle and sometimes a "flapping angle endurance limit" is defined as a limit below which fatigue damage will not occur. Certain taxiing and towing operations can also place significant bending loads on rotor masts.

During taxiing operations a forward component of thrust is introduced by applying cyclic pitch to the main rotor. This results in a bending moment being applied to the rotor head and mast. According to Grainger¹³ significant bending loads can arise particularly when taxiing over soft ground. In such situations bending loads can be reduced if the pilot introduces a lift component by applying some collective pitch.

Misalignment of shafting can lead to a considerable increase in stress levels in shafts and gears. Whenever any sub-system in the powertrain is replaced or re-installed, great care has to be taken to ensure correct shaft alignment. This usually entails a check to ensure relevant vibration signals are below prescribed limits. Boeing Vertol has been involved¹⁴ in a program to develop and test all-composite shafts and couplings designed to tolerate high misalignment angles. Composite forward and aft rotor shafts have been tested. Faust et al¹⁵ describe the development of a composite shaft for engine to gearbox connection and Jones et al¹⁶ have considered the design of composite tail rotor drive shafts for the AH-64 Apache, the UH-60 Black Hawk and the CH-47 Chinook.

(c) Rotor Systems

The rotor system is considered here to comprise⁵ rotor blades and attachments, hubs, lag dampers, rotating anti-vibration features and rotating flight control system elements. The number of rotor blades varies between two and eight. The loading of rotor blades is very complex and has been discussed by De Jonge⁷ and Bousman et al¹⁷.

In vertical flight or hover (without wind) the main rotor loading is relatively static and comprises aerodynamic lift together with centrifugal and gravity forces. In horizontal flight the main rotor blade loading is cyclic with fundamental frequency equal to the rotor rotational speed. The cyclic loads arise because cyclic blade pitch is applied to provide a component of forward thrust. The amplitude of the cyclic loads is related to drag which increases with airspeed. For the main rotor system, AUW, centre of gravity position, accelerations, and control inputs, in terms of amplitude and rate, all affect the rotor loads generated.

For the tail rotor system cyclic pitch variation is not applied. Neither the AUW nor the centre of gravity position has a pronounced effect on loads. Highest loads occur during low speed manoeuvres. The amplitude and the rate of yaw input are most critical.

The high stress levels in rotor blades have been responsible for limiting the fatigue lives of many metal blades. In recent times there has been a very strong trend towards the use of fibre composite blades which exhibit markedly slower fracture propagation rates and very long fatigue lives. However composite components exhibit different modes of degradation, including delamination, disbonding and water absorption. It is important to note however that, even when composite blades are used, metal elements are commonly incorporated in the blade assembly. For Australian Service helicopters, composite blades are used in the Black Hawk and Seahawk helicopters, and all Sea King metal blades will eventually be replaced with composite types.

Blades can be attached to the hub through hinges (bearings) to form a "hinged" or "articulated" rotor. In recent times the metal hinges for fully articulated rotors have tended to be replaced with elastomeric hinges (eg. as in Black Hawk and Seahawk) for which the required motion is obtained by deformation of an elastomer. Helicopters for which some or all of the hinges are replaced with flexible elements have entered service in the last two decades. For example the Westland Lynx has a "semi-rigid" rotor for which the flap and lag hinges have been replaced with two controlled stiffness elements.

The loading on blade attachments is important and some elements frequently have fatigue-limited lives.

When rotor blade spindles make contact with droop stops (on the rotating hub assembly), during ground operations, significant stresses can be applied to these components. Thompson¹⁸ refers to a fatigue damaging condition which initially arose due to this cause in the UH-60A Black Hawk helicopters operated by the US Army. Subsequent design modifications incorporated by the manufacturer reduced the severity of this problem. The S-70A-9 Black Hawk aircraft incorporate these modifications.

For helicopters fitted with rotor brakes significant stresses can be induced in rotor blade attachments at the time of application of the brakes. The S-70A-9 Black Hawk and S-70B-2 Seahawk helicopters operated by Australian services are fitted with rotor brakes. Very high stresses can be induced in the mechanical system if a rotor blade tip accidentally makes contact with an obstacle (eg. a limb of a tree). Apart from the blade damage resulting from such a collision severe overstressing of rotor hub and transmission components can occur. For main rotor blade collisions judged as "severe" the overhaul of rotor head and main rotor gearbox would normally be undertaken.

Autorotation (descent with engine power decoupled from the rotors) can be invoked under emergency conditions, and is also practised regularly as part of crew training. Rotor speed can increase under these circumstances. Significant levels of head bending moment and vertical acceleration can also arise. Increased centrifugal stresses on rotor hinges and other rotor head components will occur with increased rotor speeds. Maltby and Hicks¹⁹ quote the limits given in Table 2 for rotor speeds in Sea King.

(d) Rotor Pitch Control System

On fixed-wing aircraft, airworthiness of flight control systems is generally enhanced by redundancy or duplication of complete systems. For rotary-wing aircraft load transfer is usually made via a single load path for the mechanical linkages, and maintaining the integrity of such linkages generally presents more difficulties than maintaining the integrity of the flight control systems in fixed-wing aircraft. For rotary-wing aircraft the control loads can be very high and certain components frequently have lives limited by fatigue loading. Fatigue failures in rotor pitch control systems have produced numerous incidents²⁰, some of which have been catastrophic.

(e) Airframe

In this report the airframe is considered to comprise the helicopter structure external to the dynamic system. There are certain similarities with the fixed-wing airframe but structural resonances can be more common. Fatigue cracking of the airframe can occur due to in-service loads. The airframe will experience low cycle loads associated with manoeuvres, gusts, landings etc. and high cycle (vibratory) loads at blade passing frequency and its harmonics. Although the high cycle load amplitude may be small compared with the low cycle load amplitude, the high cycle loads can considerably increase the rate of growth of a crack initiated by the low cycle loads.

Certain regions of the airframe may be highly stressed; lift frames in the vicinity of the MRGB and underslung load attachment points are examples. For helicopters fitted with foldable tail pylons stresses induced in the region of the pylon hinges can be significant. For those helicopters in which the main rotor mast thrust bearing, which supports the aircraft weight, is internal to the MRGB, the gearbox casing loads can be high. Cracking of main lift frames has been a problem in RN and RAF Sea King helicopters, and those of other operators. WHL claims that it can provide suitable aircraft modifications to overcome these problems.

Helicopter undercarriage loads on "hard" landings can be significant. It follows that a major aim of the undercarriage design is to provide an efficient means of energy absorption to decrease the level of shock loads transferred to the airframe and other helicopter components, including rotor system and powertrain components. Shock loads which arise during landings while the helicopter is autorotating can be very high. While such landings can occur under emergency conditions for any helicopter, they are also regularly practised in the aircrew training role in some instances. However, many helicopters (eg. Sea King) are not cleared for autorotating to the ground in the training role.

4. HELICOPTER COMPONENT FATIGUE LIFE SUBSTANTIATION

4.1 Life Estimation Methodology and Need for Reliable Estimation

The safe-life design approach, commonly used for load-bearing metallic helicopter components, is applicable only to components whose fatigue lives can be predicted. It is not capable of anticipating failures resulting from design deficiencies or from growth of manufacturing flaws or in-service damage. According to Lambert²¹, "the CAA has stated that critical parts whose failure cannot be reliably controlled by normal lifing, because failures are not related to time or operating cycles, must have a degree of damage tolerance and must be covered by health monitoring systems".

Unexpected fatigue failures sometimes occur in components which have not been designated as having fatigue limited lives. In-service loading on many components is very complex and can sometimes lead to deficiencies in detailed design and lifing processes. Such deficiencies cannot be addressed by the fatigue substantiation methods considered in this section.

To estimate the safe life of a fatigue life limited helicopter component, both the fatigue strength and the in-service loading of the component need to be considered.

Component material fatigue strength is usually defined in terms of the familiar S-N (stress versus cycles to failure) curve, arrived at from failure tests on specimens. Bristow¹ notes the lack of uniformity of S-N curve shapes used by different designers employing the same material and cites the "wide variety of S-N curve shapes for steel used in a pitch link comparative study". Once components have been designed, their fatigue strengths are normally checked where possible using full scale tests, as indicated by Thompson¹⁸. In some instances, as indicated by Krasnowski et al⁸, a "safe" curve is drawn three standard deviations below the mean (which for a Normal failure distribution gives a 1/1000 probability of a premature failure occurring). Arden and Immen²² indicate that, in a recent

initiative, the US Army "instituted a new fatigue criterion which requires a risk level of less than one structural fatigue failure in the life of the fleet". They go on to say that "the Army has acknowledged that a reliability of 0.999999 will meet the intent of this requirement". Current design practice does not permit such a level of reliability to be achieved. Bristow¹ puts the case for substantially higher fatigue strength safety margins particularly "where the design is vulnerable to wear, fretting, corrosion, loss of clamping torque and so on".

The specification of an in-service load spectrum for component lifing purposes has been the subject of much concern to the helicopter community. Both the frequency and the amplitude of loads encountered during the operational life of the helicopter components need to be considered. Spiegel⁹ indicates that differences in assumptions made in respect of the helicopter mission load spectrum and of component strength safety margins have led to life predictions which have varied by three orders of magnitude. In all cases the degree of risk needs to be considered, but this is very difficult to quantify.

Various methodologies are employed for predicting safe lives from assumed or measured load histories. Most methods involve the use of Miner's summation law²³ to quantify cumulative damage. If the load history for a particular component has a single dominant load frequency (as is the case for many critical helicopter components) the calculation of life usage is simplified. For components subject to a less regular load history, with intermediate peaks and valleys, the rainflow cycle counting method has become widely accepted. Khosrovaneh and Dowling²⁴, and Ellis²⁵ describe the rainflow counting method.

The need to define safe lives so that the risk of in-flight failure of critical components is extremely small, is widely accepted. At the same time there is concern that over-conservative replacement scheduling can significantly affect the cost of ownership. Normally the helicopter manufacturer undertakes a loads survey on a specially equipped test helicopter to relate measured loads to flight regime. With the component strength data and the transfer function between flight regime and component loads established by measurement, it remains for the aircraft usage spectrum to be defined to allow safe lives of fatigue life-limited components to be estimated and specified. Arden and Immen²², and Thompson¹⁸ discuss the fatigue life estimation process in terms of these three elements. The element which has been the source of greatest concern and controversy is the specification of the usage spectrum. The relative merits of various approaches to the definition of the usage spectrum will be examined below.

4.2 Assumed Usage Spectrum

With this approach a flight spectrum (defining proportion of time in various flight regimes) is defined according to general specifications, or via consultation between the manufacturer and pilots²², or between the manufacturer and the customer¹⁸. The definition provides an assumed usage spectrum rather than one that is determined experimentally. The assumed usage spectrum is also referred to as the "design usage" or "design mission" spectrum. The assumed usage spectrum is applied by the manufacturer for component design purposes and for estimating safe fatigue lives of helicopter components at the time helicopters are first delivered to the customer. Thompson¹⁸ provides an example of a simple usage spectrum and fatigue damage calculation.

An obvious problem with this method is the great uncertainty surrounding the assumed usage spectrum (compared with a measured usage spectrum) such that the reliability of component safe lives comes into question. There is particular concern among operators of military helicopters that the design spectrum could be at great variance with the actual usage spectrum and this no doubt has led to many programs (Sec. 5) to measure usage spectra for military helicopters. Arden and Immen²² state in respect of the fatigue lives of components in helicopters operated by the US Army that "every helicopter system in the current inventory has several components which fall well below expectations". It is well recognized that the patterns of in-service usage for a particular helicopter type can vary markedly from one operator to another.

4.3 Measured Usage Spectrum for Sample of Fleet Aircraft

With this method actual aircraft usage is measured in a sample of fleet aircraft performing normal service duties. Data gathering is normally performed for a limited period. This method corresponds to that for MSA discussed in this report. On the basis of the findings of such a program the fatigue lives of critical components are either substantiated or revised.

In its simplest form, aircraft usage involves an analysis in terms of the identified flight regimes, the proportion of total operating time for each, and the relationship between component loads and flight regime determined by the manufacturer under test flying conditions (load survey). There is no universal definition of flight regimes; they tend to be specific to helicopter manufacturers and helicopter types. One of the difficulties encountered by the MOD²⁶ with this approach has been the identification of low speed manoeuvres for which loading on some components can be quite high. Manufacturers' load surveys have been subject to criticism²² in certain instances on the basis of insufficient account being taken of load variability due to different air conditions etc. (eg. load survey measurements made in calm air conditions cannot be readily extrapolated to turbulent air conditions).

Another variant of the measurement of aircraft usage concentrates more on loads monitoring and the generation of load spectra for each type of sortie. This variant effectively corresponds to a summation of the load spectra for the various flight regimes occurring during the sortie but excludes any reliance on the identification of individual flight regimes. The overall mission load spectrum is determined by the fleet sortie mixture. If all critical loads for fatigue substantiation purposes are measured this variant effectively removes reliance on the manufacturer's load survey which relates loads to flight regime. With this variant an attempt would normally be made to identify any manoeuvres which are damaging.

A further variant of the measurement of aircraft usage is to measure loads on selected components and to make use of properly identified flight regimes as well. The load spectrum for each regime can then be compared with that generated as a result of the manufacturer's flight loads survey. The measured flight regime profile can be used to substantiate lives of those components not covered by direct load measurements in the aircraft usage measurement program, but reliance would then need to be placed on the manufacturer's transfer function between flight regime and loads for such components.

Measurement of aircraft usage according to any of the variants described above is considered to be a vast improvement on the use of an assumed or design usage spectrum. For the results of the measurement program to be valid for fleetwide application however, it is essential that a statistically reliable pattern emerges. Also, if significant fatigue life usage occurs mainly during very rare occurrences of high load levels, a pattern which can be confidently projected fleetwide may not be obtained. Such damaging loads can only be appropriately taken account of through individual aircraft monitoring methods.

A number of helicopter fatigue substantiation programs have been undertaken around the world and some of these are reviewed in Sec. 5. Most of the programs involve both flight regime and loads data recording.

It is clear that any change in operating role may require component lives to be re-examined.

4.4 Individual Aircraft Usage Monitoring

There is a growing trend to permanently install suitable recording equipment fleetwide to support comprehensive health and usage monitoring (HUM) programs for helicopters. Two major variants of individual aircraft usage monitoring will be considered here.

For the first variant flight regime usage data, and perhaps some direct stress data, would be collected across the fleet to provide aircraft usage statistics and detect any unusual profiles. Such monitoring could establish a worst case aircraft usage spectrum for a helicopter fleet and could accommodate changes in aircraft operating role which could occur throughout the life of the aircraft. One of the major limitations of permanently installed equipment is that, for maintainability reasons, significantly fewer direct load parameters could be measured than for systems installed in a sample of fleet aircraft for a limited period. Maintainability of direct load measuring sensors and associated signal transmission circuits on rotating components would pose severe problems. Gunsallus^{27,28} describes a method of generating complete time histories of rotating component loads from loads measured on non-rotating components using a transfer matrix validated via flight tests. Such a technique could make direct load measurement a more viable option for use in individual aircraft usage monitoring. Bristow Helicopters²⁰ refers to the measurement of control load exceedances in an advanced HUM demonstrator trial involving five Super Puma helicopters.

A second variant involves fleetwide monitoring of the fatigue life usage of individual components. Technology for this variant is still emerging. Continuous measurement of parameters, from which relevant loads can be deduced directly or derived from known transfer functions, is required. Either in-flight or off-aircraft data processing could be considered. In the former case, fatigue life usage algorithms would be incorporated in the airborne computer program. The methods considered in this paragraph base component replacement lives on the actual load history and therefore allow optimum lives to be achieved without compromising safety of operation. Harrington et al²⁹ describe a concept demonstration of advanced techniques for use in helicopter component fatigue life usage determination.

LCF life usage monitoring of engines in individual aircraft is now provided for some engines. The GE T700 series engines in Black Hawk, Seahawk and other helicopters are fitted with an engine-mounted history recorder which tracks engine usage cycles. However it is understood that, at this time, most operators of these engines employ a maintenance schedule which defines safe lives in operating hours. Prototype equipment suitable for in-flight tracking of helicopter gear fatigue life usage is described by Fraser and King³⁰.

While there is potential to include fatigue life usage accumulation within HUM systems, further development is required before it could be implemented. HUM systems are still relatively immature, with most current activity being focussed on concept evaluation trials.

5. COMPONENT FATIGUE LIFE SUBSTANTIATION PROGRAM REVIEW

5.1 General Comments

A summary of some current and recent helicopter component fatigue life substantiation programs is provided in Table 3. All of these programs involve the recording of parameters to assist in the identification of the aircraft flight regime or manoeuvre pattern. Most programs also involve the recording of actual loads or stresses in selected components. The recording of actual loads or stresses provides some degree of verification of loads measured in manufacturer's load surveys. The reliability of such surveys has been of concern to operators. In all the fatigue substantiation programs, referred to in Table 3, the helicopter manufacturer is involved in the translation of measured flight and loads data into the fatigue life usage of selected components.

The number of overseas helicopter operators (particularly of military helicopters) undertaking fatigue substantiation programs at or near the present time, lends credence to the argument that the RAAF should undertake a similar program for the S-70A-9 Black Hawk. Configuration and operational environment differences between the Australian and US Army Black Hawk aircraft would limit the extent to which US program (Table 3) findings could be read across to the Australian fleet.

Some of the programs in which helicopter operators in the UK and USA have been involved, or are planning to be involved in the near future, are examined briefly in the following sub-sections.

5.2 UK Military Helicopter Programs

During the past decade the MOD has been engaged in a range of monitoring programs^{26,31} aimed at the substantiation of helicopter component fatigue lives.

(a) Sea King

The Sea King helicopter, in service with the RN and RAF, has been the lead helicopter in defining methodology and instrumentation for the fatigue substantiation work.

During 1980/81 about 150 hours of flight data involving 134 sorties were recorded in a Manoeuvre Recording program applied to a RN Mk 2 Sea King operating in the training role. The aim of this program was to define the way in which the aircraft was used. The data collection philosophy adopted was to measure control inputs and aircraft responses, collectively referred to as parametric data. From these data it was hoped that specific manoeuvres and flight conditions could be identified. The data were analysed by RAE but great difficulty was experienced in automatically recognizing all flight manoeuvres. A separate but related program referred to as the Phase 1 Sea King Loads Survey followed. In this program, which was completed in 1984, 22 hours of flight and loads data covering a full range of operational sortie types were collected in a fully instrumented development aircraft. It was concluded that automatic recognition of all manoeuvres, particularly the low speed ones, would be difficult. Low speed manoeuvres are important in that high levels of tail driveshaft torque are likely. The Loads Survey program showed that some flight conditions revealed higher loads than were used in the original component life substantiation, although the differences were not sufficient to raise major concerns over component lives. It was concluded that any future programs to substantiate component fatigue lives should include some direct stress measurements. A modified version of the compact Plessey EULMS logging equipment (which includes a Vinten quick-access cassette recorder) was used in the Manoeuvre Recording program and a much larger Plessey MODAS trials installation was used for the Loads Survey.

An ad hoc HODR program on a RAF Mk 3 Sea King operating in the Falkland Islands was undertaken in 1984. Twenty three analogue parameters, including some providing direct stress measurements, and four digital events were recorded³². Data from more than 100 flights were analysed by the RAE which successfully produced cumulative fatigue damage plots and was able to identify most manoeuvres. The Plessey EULMS installation used for the Manoeuvre Recording program was transferred to the Falkland aircraft for this program.

A major HODR program in which more direct stress/load measurements will be made than in the above ad hoc program, is being undertaken on five selected Sea King aircraft covering all versions of this helicopter in UK military service. Four RN and one RAF aircraft have been selected for this program which is currently under way. This program, which is examined more closely in Sec. 7 involves the recording of 55 analogue channels and 3 digital states. WHL is the prime contractor involved in the measurement and the analysis of the HODR data. Rolls Royce will analyse the engine data. A variant of the Plessey EULMS equipment used previously is being used for this program.

(b) Chinook

A Chinook HODR program commenced in 1985 on RAF aircraft. Originally three aircraft were involved to cover the UK, Germany and Falklands operating theatres, but the Falklands aircraft was lost. This program involved 33 analogue channels and 9 discrete states. Five direct stress measurements were included. Components for which fatigue life revalidation was considered particularly important included the aft vertical rotor mast and other major components in the powertrain, engines, rotor control mechanism and airframe. The "aft vertical link" which forms part of the aft rotor pitch control mechanism, is considered to be a life-critical item and strain gauge measurements were taken for this item. Forward, centre and aft cargo hook loads were also measured via

strain gauges. Not all the data gathered were analysed initially but are available for later use if required. For example engine parameters N1 (gas generator speed), N2 (power turbine speed), P3 (compressor delivery pressure), PTIT (power turbine inlet temperature) and TORQ (engine torque) were recorded but only N1 was used for engine LCF computations. Another aim of the program was to identify flight profiles (number of engine start/stops and landings, hover periods etc.). The airborne instrumentation for the program was supplied by Boeing Vertol and comprised a Plessey SUMS (similar to EULMS). Under MOD contract, Plessey Avionics transcribed the flight tapes and analysed the engine data according to algorithms supplied by Lycoming.

(c) Lynx

Six aircraft are to be used in the Lynx HODR program, two from the RN, two from the RAF and two from the Army. This program is scheduled to commence about six months after the start of the five-aircraft Sea King HODR program.

(d) EH101

There are no plans to extend the HODR program to other service helicopters. Fatigue life usage monitoring techniques will be developed as an integrated function of the health and usage monitoring system for the EH101 helicopter. The system will use the Aircraft Management Computer for airborne data processing and storage.

5.3 UK Civil Helicopter Programs

(a) Chinook

A program similar to the MOD HODR programs has been funded on a British Airways BV.234 Chinook helicopter operating over the North Sea. The first stage, which involved the recording of simple parameters such as speed, altitude, engine torque etc., commenced in 1984. It is proposed that later stages will include measurement of loads in certain critical components.

(b) Super Puma

A program which involved the measurement of flight regime recognition parameters on the Super Puma helicopter was completed by Bristow in 1987. No direct stress measurements were made. Component fatigue life substantiation was performed by Aerospatiale using component strength data and previously derived transfer functions between flight regime and component loads.

5.4 USA Military Helicopter Programs

Military operators in the USA have undertaken (or propose to undertake) a number of helicopter fatigue substantiation programs which involve loads monitoring. Some of the programs of which the author is aware are examined briefly below.

(a) UH-60A Black Hawk

Under US Army contract, Sikorsky is undertaking a "structural" usage monitoring program on the UH-60A Black Hawk aircraft. Selected load parameters will be monitored in the powertrain, rotor system and flight control mechanism. A measurement program of six months duration was scheduled for two Army aircraft in late 1989, but this program has been put on-hold.

The program makes use of the Structural Usage Monitor (SUM) airborne data system supplied by Canadian Marconi (CM). However the system has been further developed and modified by Sikorsky for this application. The on-board SUM system will validate the mission load spectrum against which the "structural" adequacy of the aircraft was originally based. In-flight recognition of approximately 200 flight regimes will be provided. Load spectra (proportion of total flying time spent in each of 16 load bands) corresponding to each flight regime will be generated. Life substantiation for the main rotor shaft and flight control linkages will be included in this program. Time history tables of input data at the time of a load occurrence are generated in compressed format. Discrete information such as the number of take-offs, the number of rotor starts etc. are tabulated. The SUM system flight data will be downloaded daily onto disks in ground support equipment (GSE) which is periodically taken on-board the aircraft by ground staff. The on-board system can retain data from multiple flights in solid state digital storage.

(b) SH-60B Seahawk

Under US Navy contract, Sikorsky is undertaking a monitoring program on the SH-60B Seahawk aircraft with similar aims to that detailed above for the UH-60A SUM program. A data system similar to that being used in the UH-60A SUM program will be employed. Six aircraft will be involved in this program, with some operating from frigates and some from carriers. Extra parameters relative to the US Army SUM program will be recorded. It is expected that data collection for this program will take place in 1991.

(c) OH-58C Kiowa

The US Army, with the support of Bell Helicopter Textron Incorporated, is undertaking a component fatigue substantiation program in relation to its OH-58C helicopter. The program, described in more detail by Buckner³³, is aimed at measuring the flight spectrum which will reflect some new operating missions recently introduced for the aircraft. The measured spectrum will be compared with the design mission spectrum. Thirteen parameters for flight regime recognition are measured. No direct load measurements are included.

(d) MCH-53E Super Stallion

A preliminary investigation for a usage monitoring program for the Super Stallion helicopter has been undertaken by Sikorsky under US Navy contract. The aim of this investigation was to design, manufacture and bench-test a system which could accumulate exceedance data, recognize flight regimes, and categorize and compress data for storage. This work was completed in 1987.

(e) AH-64A Apache

Under US Army contract MDHC is demonstrating an Enhanced Diagnostic System (EDS)²⁹ in one Apache. One of the primary functions of the EDS is to monitor loads and aircraft usage. A Lear Siegler airborne data system interfaced with the aircraft 1553 bus is used. An on-aircraft display of engine health status, exceedances, system fault status and on-board diagnostic results is provided. Expert system logic is being implemented to demonstrate the display of caution/warning/advisory information through the analysis of the multiplexed bus data. An on-board cartridge recorder stores data for post flight analysis with a unit level computer. Flight regimes are recognized, and their severity and duration are monitored. Time-history data for usage monitoring are acquired. For determining load histories some loads are deduced from a generalized transfer function relating loads to flight regime, and some loads are measured directly. In the former case the transfer function coefficients are determined empirically.

6. DATA REQUIRED FOR FATIGUE SUBSTANTIATION

6.1 General Comments

Normally the helicopter manufacturer would be contracted to support the substantiation of component fatigue lives from flight regime recognition data and selected loads measured in the operator's aircraft while it is performing normal duties. The manufacturer would have the required component strength data derived from stress analyses supported by full scale component life tests. Furthermore the manufacturer would be familiar with component lifing methodology and would be favourably placed to undertake component fatigue life substantiation.

The types of data required to be acquired and collated in such a program will be considered in this section.

6.2 Fleet Utilization Data

Although a fatigue substantiation program requires the automatic collection of data in a sample of fleet helicopters there are certain data, often logged in fleet records, which are relevant to such a program. Such data may include sortie type, AUW, flight duration etc. An example of some sortie types and proportion of fleet time for each is illustrated in Table 4 for a particular fleet of helicopters.

At the completion of the data collection program the sortie distribution for the instrumented aircraft could be compared with that obtained from the broader based fleet records to see whether the operations for the aircraft involved in the measurement program were truly representative of fleet operations. If mission loads are related to sortie types the results could be readily factored, if necessary, to make them applicable to the fleet operations.

6.3 Configuration Details

AUW and position of COG are important parameters which are difficult to log automatically.

In the US Army and US Navy SUM programs (Sec. 5.4) Sikorsky plans to record a 5 second time history of certain parameters to enable AUW to be estimated for each flight (at an identified time during the flight). It is proposed that AUW will be estimated at the time a detailed analysis of the collected data is undertaken. The method is based on the relationship between power requirement and gross weight corrected for certain conditions. AUW will vary considerably throughout a long flight as fuel is consumed, and could typically vary by about 25% for a long flight.

It is standard practice to calculate or estimate COG position prior to take-off to check that it is within prescribed limits. The net moment about the nominal COG position of non-standard items (including personnel) installed for the flight is calculated; the net moment divided by AUW gives the shift in COG position. For example, the prescribed COG limits for the Sea King helicopter are 9.4 inch forward to 8.6 inch aft of the datum position. Sometimes a verification that the COG is within limits is performed qualitatively by the pilots based on the "feel" of the controls at lift-off.

In the absence of automatic logging methods, it would be desirable to log AUW (take-off value) and COG manually, if possible.

6.4 Flight Regime Recognition Data

A major requirement of a helicopter fatigue substantiation program is to collect data from which the flight regime may be recognized. For comparison with the assumed mission lifting spectrum (Sec. 4.2), it is desirable that the defined regimes correspond with those adopted for the assumed mission spectrum. A uniformity of definition of flight regimes does not exist between helicopter manufacturers. For Black Hawk, Sikorsky defines about 200 regimes covering various ranges of manoeuvre and steady flight.

The identification of various flight regimes from recorded aircraft control and response data is by no means straight forward. Although a flight regime is a more simple entity to recognize than a manoeuvre, it is significant that Holford³² experienced great difficulty in consistently identifying specific manoeuvres from measurements made on a Sea King helicopter engaged in the training role during 1980. She indicated that the problems were accentuated for low speed manoeuvre flight.

Under US Army contract, Sikorsky has developed methods for in-flight recognition of flight regime. It is proposed that these methods be demonstrated in the Structural Usage Monitor (SUM) programs (Sec. 5.4) being undertaken for the UH-60A Black Hawk and SH-60B helicopters.

6.5 Loads Inferred by Indirect Measurement

It may be impractical to directly measure all loads which may be of fatigue significance. Deductions of some component loads (considered important for component fatigue life substantiation purposes) from other measured parameters such as those which define flight regime, and from a knowledge of helicopter dynamics, may allow reasonable estimates to be made.

In programs for which fatigue substantiation of engine components is required, it is normal for the engine manufacturer to implement algorithms which deduce LCF and thermal fatigue life usage from measurements of gas generator speed and engine temperature excursions.

6.6 Directly Measured Loads

Where feasible it is desirable that some direct measurements of critical loads be made. Holford³² outlines the philosophy of direct load measurement and fatigue assessment for component fatigue substantiation programs undertaken in respect of UK military helicopters. Measurements on rotating components, such as rotor systems, are complicated because of the need to convey the signals to stationary circuits. Slip ring assemblies have been traditionally used in this role in the past but maintainability can be a major problem. A number of non-contacting transmitter-receiver systems, such as those using radio or optical communication, are now being used for applications such as this.

Normal loading on helicopter rotating machinery and airframe components tends to be cyclic in nature, the variations being at the fundamental main rotor frequency (typically about 3 Hz) or higher multiples of main rotor frequency depending on where the component is located. From a fatigue damage estimation point of view, life usage can be estimated if the time history of the mean stress and the amplitude of the "oscillatory" component of stress is known in addition to the frequency of application. Fatigue damage³² varies as the third or fourth power of the load cycle amplitude and is roughly linear with the mean load. In many cases the mean load component is ignored. Holford states that "when substantial mean load variations do occur between manoeuvre states, it is necessary to reconstitute a single load history from the mean and vibratory signals from which loading cycles, which comprise a peak and a trough from different manoeuvres or conditions, may be identified". Loading cycles whose amplitudes fall below the fatigue endurance limit can be ignored. For helicopters the rotor frequency is very nearly constant and the frequency of the stress variation can normally be inferred. The rotor frequency would normally be measured. Direct recording of the full stress waveform over the complete load cycle would place prohibitive demands for data storage when the required number of such direct load channels is considered. The recording of the load cycle amplitude allows much lower sampling rates and more compact storage systems to be used. With this technique the rate of change of amplitude becomes the important parameter to be considered when setting the minimum sampling rate. If measurements of mean load values are also required, separate channels need to be allocated for their recording.

7. PARAMETER LIST

7.1 General Comments

The full list of parameters which should be measured in a fatigue substantiation program will differ from one helicopter type to another and may vary according to operational duties for a given helicopter type. However most of the parameters required for flight regime recognition, and many of those for powertrain and engine monitoring (if required) in support of such a program, will tend to follow a fairly standard pattern. Loads monitoring in rotor systems, pitch control systems and airframe will generally require the

special fitting of strain sensors and possibly load cells at selected locations. Strain gauges have been traditionally used for strain measurement but fibre optic sensors are currently being assessed³⁴ for use in such applications. The latter provide promise for improved durability in more severe environments.

Accurate monitoring of load/stress levels is desirable as life usage typically increases dramatically and non-linearly for stresses in excess of the endurance limit. The conditioning of some parameters for display on cockpit instruments can introduce inaccuracies which may be avoided in loads monitoring programs by intercepting signals at points ahead of the standard aircraft conditioners and employing more accurate conditioning systems.

A representative set of parameters³⁵ proposed to be measured in the five-aircraft MOD Sea King HODR program (Sec. 5.2) is given in Table 5 together with initially defined sampling rates. A list of parameters to be measured in the US Army and US Navy SUM programs (Sec. 5.4) for the UH-60A Black Hawk and the SH-60B Seahawk is given in Tables 6 and 7 respectively. A provisional list of parameters proposed for the Australian S-70A-9 Black Hawk fatigue substantiation program is given in Table 8. In each case, some parameters will be derived from sensors which are included in the standard aircraft installation and some will require the fitting of special sensors.

Some comments on the selection and measurement of parameters for flight regime recognition and component loads measurements are provided in the following subsections.

7.2 Flight State Recognition Parameters

A typical selection of flight control inputs and aircraft response parameters is given in Table 6. For the SUM programs, specially fitted rotary variable differential transformer (RVDT) sensors will provide readings of cyclic pitch lever lateral position and yaw pedal position. Outside air temperature will be measured using a specially-fitted platinum-resistance probe. Aircraft attitudes will be derived from the standard aircraft gyros. Aircraft vertical acceleration will be measured with a specially-fitted servo accelerometer. Altitude and indicated air speed (IAS) will be measured with the standard aircraft airspeed and altitude transducer, which employs capacitive pressure sensors. Sampling rates for most of these parameters will be 4 Hz. Accurate measurement of low values of airspeed is very difficult.

7.3 Engine Parameters

Measurement of parameters which can, with suitable algorithms specified by the engine manufacturer, be used to measure life usage of engine components is not always included in helicopter fatigue substantiation programs. Such measurements are not included in the US SUM programs referred to above.

Excursions (Sec. 3) in engine rotational speeds and temperatures are the major contributors to LCF usage of engine components. While it is the function of the engine control system to maintain power turbine rotational speed N_2 within close limits, some variations will occur. Gas generator rotational speed N_1 and turbine inlet temperature will

both increase with increased power demand from the engine. It is standard practice to include N_1 and N_2 sensors to allow values to be displayed on cockpit meters. Sensors for N_1 and N_2 often provide analogue pulse rate (or frequency) signals.

For engine component LCF life substantiation in helicopters, at least N_1 and turbine entry temperature $T_{4.5}$ would have to be measured. For the Sea King HODR program (Table 5), compressor delivery air pressure is also monitored. Fuel mass flow is also to be measured in that program by measuring the volumetric flow of fuel to each engine via turbine flowmeters together with the temperature of the fuel.

Engine inertia does not permit rapid fluctuation of engine parameters and a sampling rate of 2 Hz should be adequate.

7.4 Powertrain Parameters

Torque is the load parameter of major interest for the powertrain. It is standard practice to sense and display engine torque on cockpit torquemeters. Engine torque has traditionally been a very difficult parameter to sense accurately.

In earlier helicopter designs it was more common to sense torque external to the engine, such as in the MRGB. Hydraulic sensing such as that used in Sea King has been a common method. In that helicopter helical gear axial loading, which is proportional to applied torque, is supported by a "cushion" of oil the pressure of which is proportional to torque. The hydraulic systems are not renowned for high accuracy, with 5 to 10% being typical figures claimed. The estimated fatigue life usage of powertrain components can be dramatically affected by torque measurement inaccuracy.

It is now becoming a widely accepted practice to incorporate torque sensing as an integral element of helicopter turboshaft engine design. For example, such sensing is incorporated in the GE T700 and the RTM 322 series engines. Versions of the T700 engine are installed in Black Hawk and Seahawk helicopters operated in the USA and Australia. For the T700 engine, the relative twist between an unstrained inner shaft and the power turbine shaft is measured. The sensor comprises magnetic tabs (two on the power turbine shaft and two placed at the free end of the inner shaft) and a stationary pick-up coil. Relative phasing of the detected signals provides a measurement of the power turbine shaft twist which is proportional to transmitted torque. GE representatives indicated that the engine torque sensor would have a maximum inaccuracy of about 2% of rated engine torque, but said "the inaccuracy of the cockpit torque indicator could be significantly greater" because of further inaccuracies introduced in the signal conditioning and indicating system.

To completely define the torque transmitted through all powertrain components (Sec. 3), the tail driveshaft torque must also be known. Means for tail driveshaft torque measurement and cockpit display are not normally provided by permanently installed equipment. A tail driveshaft torque sensor comprising a specially fitted strain gauge and a radio frequency FM telemetry transmitter, manufactured by EEL, is to be used in the MOD Sea King HODR program. Because of the difficulty of measuring tail driveshaft torque it is frequently omitted from the load parameter list. Where tail driveshaft torque is required for fatigue substantiation of tail drive components and no direct torque

measurement is made, the value of tail driveshaft torque would have to be estimated from measured engine torque and aircraft flight regime, which can be difficult since problems arise in recognizing low speed manoeuvres (Sec. 5.2) for which high levels of tail driveshaft torque are likely.

A 4 Hz sampling rate should be adequate for powertrain torque measurement.

Stresses, other than those arising due to applied torque, for rotor masts (which are normally integral with rotor gearboxes) will be considered in the next section.

7.5 Rotor System Parameters

Some direct stress measurements on rotor system components using strain gauges are usually desirable. Because the strain gauges need to be placed on rotating components the transfer of the signals to stationary pick-up circuits is a major problem.

The measurement of blade stresses can be extremely troublesome as it is difficult to maintain strain gauges, particularly outboard ones, in a serviceable condition. The measurement of stresses in composite blades is not considered to be warranted as these blades are not considered to be fatigue-critical. Blade stresses are not being measured in the MOD Sea King HODR program or the SUM programs.

Bending stresses in main rotor masts, due to cyclic loading at main rotor shaft frequency under cyclic pitch conditions, are frequently significant. Such bending stresses are measured in most helicopter fatigue substantiation programs. Stresses in other main and tail rotor hub and rotating flight control components, are to be measured in the Sea King HODR program (Table 5). Once again strain gauges and EEL FM telemetry transmitters are to be used.

Provided the required stresses are converted to amplitude and mean values (or peak and trough values), a sampling rate of 4 Hz would be suitable for rotor component stress measurement.

The amplitude of bending stresses in main rotor masts is proportional to the amplitude of the cyclic blade flapping angle, and measurement of flapping angle can be considered as an alternative to the measurement of main rotor mast bending stress. ARL has been involved in the measurement of main rotor flapping angle in a RAN Sea King helicopter using a potentiometric sensor insert in the metallic pitch control hinge pin for a trials installation. It is much more difficult to measure flapping angle directly in helicopters fitted with elastomeric hinges (eg. Black Hawk).

Under autorotation conditions, with engine power decoupled from the drive system, rotor rotational speed cannot be deduced from engine power turbine rotational speed N_2 . As centrifugal loads in rotor systems can increase significantly with increased rotor speeds during autorotation, it is essential that main rotor rotational speed N_1 be measured.

7.6 Rotor Pitch Control System Parameters

Loads in flight control system components can be measured with the aid of specially fitted strain gauges. In the MOD Chinook HODR program strain in the flight control system aft fixed link, which is fatigue life limited, was measured. In the MOD Sea King HODR program strain is measured in the tail rotor bell crank lever. Loads in control links are to be measured in the US Army and US Navy H-60 series helicopter SUM programs (Tables 6 and 7) and are proposed to be measured in the Australian Black Hawk MSA program (Table 8). A sampling rate of 16 Hz is to be used in the US programs.

Care needs to be taken with electrical connections to strain gauges fitted to elements of the flight control system, as these elements move relative to the airframe.

7.7 Airframe Parameters

Some areas of the airframe may be highly stressed and some stress measurements may be warranted. For the MOD Sea King HODR program, the mean stress and the amplitude of the vibratory stress in the main lift frames and the tail pylon hinge are measured via strain gauges. The underslung load is measured via a load cell placed in the cable to the hook. Similar load measurements were made for the forward, centre and aft hooks in the Chinook HODR program. Undercarriage landing loads (Table 7) are to be measured in the US Navy SH-60B SUM program. The sampling rate required for airframe parameters can be fairly high; a rate of 32 Hz is used for the undercarriage loads referred to above.

7.8 Other Parameters

Other information required for fatigue evaluation purposes relates to cycles which occur once per rotor engagement or once per flight. A flight is defined as a rotor engagement, taxi, lift-off, airborne manoeuvres, landing and rotor shut-down. A lift-off and landing cycle is called a Ground-Air-Ground (GAG) cycle, and can occur with or without a rotor start/stop cycle. Rotor accelerations to 100% rotor speed and back down again have significant effect on the fatigue life usage of some components. Absolute maximum and minimum values of loads experienced during flight are also important for fatigue analysis purposes. The number of GAG cycles, the number of rotor starts and extreme values of loads for each GAG cycle would normally be required.

Lift-off can be regarded as a discrete parameter which can be tracked with a weight-on-wheels sensor switch. Similarly rotor engagement would be detected as a discrete switch state. Other discrete values corresponding to such states as rotor brake application, automatic flight control system (AFCS) engagement and external load application, may need to be recorded in support of the helicopter fatigue substantiation program. The discrete parameters can be most effectively handled as one-bit digital inputs. A sampling rate of 1 to 4 Hz would be adequate for the discrete parameters.

Other parameters, such as those discussed in Secs. 6.2 and 6.3, may need to be tracked manually.

8. FUNCTIONAL REQUIREMENTS OF DATA SYSTEM

8.1 General Requirements

The airborne data system for use in helicopter fatigue substantiation applications must be capable of acquiring data over a considerable period. Normally data collection would be continued until the mission flight and loads spectra have stabilized. One operator, who has been involved in helicopter component fatigue substantiation programs, has indicated that the collection of about 200 hours of "good" in-flight data per aircraft may typically be required. To take account of the problems associated with the collection of good data, the actual data collection period would need to be suitably increased. One major organization which has been involved in helicopter component fatigue substantiation for a considerable period has suggested a 65% increase may be realistic. Data are to be collected while the aircraft are involved in normal duties, and should encompass the range of normal sorties for the specified aircraft fleet. Collection of data for the required range of sorties is often achieved by making measurements on a number of aircraft engaged in different operating roles.

The airborne systems must be robust and reliable. However fleet operations would not be affected by short-term unserviceability of the airborne systems as the program involves the collection of representative samples of data only. The airborne data system environmental specification need not be as severe as that required of permanently installed systems certified for fleetwide use. However a good environmental specification is essential to enable the equipment to cope with the operational environment and ease maintainability problems.

The helicopter fatigue substantiation application usually requires that the data system elements be of low mass and volume, so that their presence does not interfere with normal duties being performed by the aircraft and aircrew.

A general arrangement of the data system elements required for helicopter MSA is depicted in Fig. 2. Functional requirements for these elements, especially those which apply to the S-70A-9 MSA program, will be considered in the following sub-sections.

8.2 Input Parameter Sensing

The specific input parameters selected for measurement in a component fatigue substantiation program will vary from one helicopter type to another. Typical parameter and sensor requirements have been examined in Sec. 7. For the Australian S-70A-9 Black Hawk MSA program, it is proposed that about 23 analogue and 3 discrete inputs will be measured (Table 8). The provisional list of analogue channels comprise 13 which are derived from the standard aircraft system and 10 which require specially fitted sensors.

8.3 Signal Conditioning

Suitable conditioning of input signals will be required either within the programmable signal acquisition unit (Sec. 8.4) or within an external signal conditioning unit. Most of the sensors generate analogue voltage outputs, but analogue phase and analogue frequency outputs are usually generated by some sensors. Aircraft attitude parameters are

normally available as synchro signals which require special conditioners to convert the analogue phase information to analogue voltage signals (or directly to digital form). The rotor speed parameter, which is usually sensed as an analogue frequency signal, needs to be converted to an analogue voltage signal or directly to digital form.

Typical requirements for the analogue voltage signals are amplification, level shifting, AC to DC conversion and filtering. In cases where both the vibratory amplitude and the mean value of a signal is required, conversion to two DC outputs, either an amplitude and a mean value, or a positive and a negative peak value, will be necessary. Many of the stress/load measurements required for component fatigue substantiation purposes are in this category. Low pass filtering, at typically three times the signal bandwidth, is required to reduce extraneous noise effects. Special signal conditioning requirements for strain gauge load sensing bridges include the provision of bridge excitation and, normally, the facility to temporarily introduce shunt calibration resistors.

8.4 Programmable Signal Acquisition

The programmable signal acquisition unit would be expected to:

- (a) Set the sampling sequence and individual channel sampling rate for the input parameters.
- (b) Multiplex the analogue channels and convert the outputs to digital form.
- (c) Provide microprocessor control of data flow and, preferably, perform some data compression and/or processing during flight.
- (d) Store appropriate data on-board for later transfer to a ground system.

The extent of on-board data compression and processing is a most important consideration as this will affect the storage requirement and the amount of ground station processing required. For the MOD HODR programs (Sec. 5.2) virtually no on-board processing is done whereas extensive on-board processing is included in the SUM programs (Sec. 5.4) being undertaken by Sikorsky under US Army and US Navy contracts. The on-board processing functions performed by the SUM systems are examined in more detail in Sec. 8.9.1.

It is important that any on board processing proposed be achievable within the constraints imposed by the requirement to operate in real time.

The input parameter sampling rates need normally to be set high enough to follow the changes in parameter values. However, as mentioned earlier, it has become common practice to convert load amplitude values to a DC value, in cases where the frequency of the load variation is directly proportional to rotor speed. For such parameters the rate of change of signal amplitude becomes the important characteristic for setting the sampling rate. To prevent aliasing errors the sampling rate used for any given parameter must be at least twice the highest signal frequency present in the input to the multiplexer. The Sea King HODR program (Table 5) can be satisfied with an overall sampling rate of 256 Hz. A nominal 256 Hz overall sampling rate should be adequate for the US Army UH-60A

Black Hawk SUM program (Table 6) and for the Australian S-70A-9 Black Hawk MSA program (Table 8). However a nominal 512 Hz overall sampling rate is required for the US Navy SH-60B SUM program, dictated by the additional parameters to be measured in that program (Table 7).

8.5 Manual Input

Although manual entry of certain information for storage together with measured data would be an advantage, a requirement to enter such information may be unacceptable in the operational environment in which the data are to be collected.

In situations where entry of such data is permitted, it would be advantageous to record such information as sortie type, aircraft configuration details, AUW and COG position. If date/time information is not automatically encoded with the measured data, it would be advantageous to also enter these data via a cockpit unit. A facility to enter and store such data as aircraft tail number would be useful. In the latter case non-volatile storage would have to be provided so that re-entry of the data is not required for following flights. It is important that any cockpit data entry facilities be simple and rapidly implemented.

If provision were not made to encode the above data, it would be desirable for them to be recorded manually on paper (if the information is available) and arrangements be made for them to be entered into the ground station records.

8.6 Airborne Data Storage

The amount of airborne data storage required will be greatly affected by the extent of in-flight data processing and reduction performed. If most data processing is performed off the aircraft, the amount of storage required will be a maximum. For the UK HODR programs the approach is to perform the data processing off the aircraft and for the US SUM programs the approach is to perform most of the required data processing during flight. Since the provision of adequate storage for processed data is not considered to be a very demanding requirement, only the case where all the data are processed at the ground station will be examined below.

The duration of the longest flight will set an initial minimum limit for the amount of airborne data storage required. The Sea King HODR program (Table 5) can be satisfied with a 2.25 hour storage duration at an overall sampling rate of 256 Hz. The normally configured S-70A-9 Black Hawk helicopter has an endurance of about 2.5 hours. However, with external fuel tanks fitted for ferrying applications, the endurance can be increased to about 10.2 hours. Hence storage for at least 10.2 hours of flight data must be provided for the S-70A-9 Black Hawk.

The amount of storage required per flying hour will vary according to the type of data compression employed. Based on a 256 Hz sampling frequency the following approximate amounts of in-flight storage would be required:

- (a) Most extravagant storage technique using 2 bytes per parameter measurement: 1.8 Mbyte per flying hour.

- (b) Data compacted to 1.5 bytes per parameter measurement but not compressed: 1.4 Mbyte per flying hour.
- (c) Data compressed in form where full recovery is possible: 1.0 Mbyte per flying hour (estimated).

If data compression as for (c) were provided for the S-70A-9 MSA application, a minimum of 10 Mbyte of in-flight storage would be required. If a form of data compression, for which data are discarded during flight, were used less data would need to be stored per flying hour.

In situations where it is not convenient to change the data storage medium or to transcribe the stored data via a suitable data extraction unit after each flight, the provision of storage to cover multiple flights is desirable. For the Australian Black Hawk it is not uncommon for "hot re-fuelling" to be done and for the aircraft to take off again without engine shut-down. Extraction of data during such a landing is not permitted.

Although maximizing the amount of in-flight data processing can overcome problems in storing sufficient data to cover an adequate duration of flying time, it does introduce some disadvantages. For example the collation of data according to sortie type would not be possible, if the collated data were not separated on a flight-by-flight basis. Data verification is also much more difficult if the data are extracted only in highly processed form. The retention of "raw" data can have advantages if there is a need to look at the data in more depth for scientific or other purposes at a later time.

A requirement to store a large amount of data will introduce a mass penalty for the on-board data system, but the ever increasing solid state digital storage capacity per unit volume can reduce the penalty to a minimum. The compromises associated with in-flight versus ground station data processing and collation have no doubt contributed to different approaches being adopted in helicopter fatigue substantiation programs currently being undertaken around the world.

8.7 Flight Line Test

While maximum use of built-in-test (BIT) capability within the airborne data system is very desirable, it is essential that a facility be provided for functionally testing the airborne system at the flight line. Selection of any desired channel and digital display of the encoded digital word for that channel would be required.

8.8 Data Recovery and Verification

It is likely that the data stored during flight would be transferred to a PC ground station computing system, probably located at the operating base for the aircraft. Any special reading devices for the stored data would need to be interfaced to the computing facility.

Some intelligent screening of the data would be required, either by the airborne system or by the ground system, to flag, reject and take account of the time period associated with bad data. Some bad data are likely to be found in this application in the light of the severe operating environment.

8.9 Data Collation and Analysis

The specific requirements for data collation will vary from one application to another. Since helicopter manufacturer involvement will normally be essential for the component fatigue substantiation process, it is important that data be collated in a form which matches that adopted by the manufacturer. Approaches used by Sikorsky and WHL in current helicopter fatigue substantiation programs are examined below.

8.9.1 Sikorsky Data Collation Requirement

The form in which the data are to be collated for the US Army UH-60A Black Hawk and the US Navy SH-60B Seahawk SUM programs has been defined by CMC³⁶ and Sikorsky³⁷. Data collation for the SUM programs is performed during flight by the signal acquisition unit (Sec. 8.4). A similar data collation requirement is presumed to apply for the Australian S-70A-9 Black Hawk MSA program, but without excluding data collation at the ground station as an option.

Sikorsky identifies flight regimes during flight at a 1 Hz rate. Identification of flight regimes at a repetition interval of 1 second of flying time is a presumed requirement for the S-70A-9 MSA program.

The total time spent in each flight regime and the number of times each regime is entered is required to be stored in a flight regime time and occurrence table. For the 200 regimes defined this table will require 400 entries.

For each regime, a histogram containing the cumulative time spent in each of 16 bands for each measured load parameter (except main rotor droop stop load) is required to be generated for the US Army SUM program and is presumed to be required for the S-70A-9 MSA program. Only one histogram is required for main rotor droop stop load. For the 6 load parameters defined (Table 8), storage for 16016 entries ($5 \times 200 \times 16 + 16$) is required.

The number of GAG cycles, the number of rotor starts and extreme values of loads for each GAG cycle are required.

For each of the 17 parameters used for flight regime recognition a histogram, which contains the cumulative time spent in each of 16-bands for each parameter, is required. Storage for 272 entries is thus required.

Sikorsky also requires parameter time history tables to be generated for estimating AUW, for obtaining a snapshot associated with load exceedances and for tagging discrete parameter changes with clock time. A 5 second time history table for engine torque, airspeed, altitude and outside air temperature is required for the AUW estimation. Storage required for the time history data is difficult to estimate; unlike the histogram data they are not progressively overwritten.

8.9.2 WHL Data Collation Requirement

WHL³⁵ has defined the data collation and analysis requirements for the Sea King HODR program. For that program the acquired data are to be collated and analysed by WHL using ground based equipment. The form of data collation required is detailed below.

(a) Event Durations

The durations of the following events are to be logged:

- Mission time (time for which rotor brake is released).
- Airborne time (time for which undercarriage scissors switch is off).
- Total time for which AFCS is engaged.
- Duration of each autorotation.

(b) Event Counts

The following events are to be counted:

- Take-off/landing.
- Rotor start/stop.
- Autorotations.

(c) Parameter - Time Spectra

Totalized times in various level bands of a particular parameter (two-dimensional spectrum) or of a combination of two parameters (three-dimensional spectrum) are required as listed below (with a "/" designating that a three dimensional spectrum is required):

- Airspeed/density altitude (calculated from pressure altitude and outside air temperature).
- Main rotor speed.
- No. 1 engine torque, No. 2 engine torque, sum of engine torques, tail rotor torque / mean load in tail bellcrank lever, tail rotor torque / vibratory bending load in tail spider arm and main rotor torque (rationalized total torque minus tail rotor torque).
- Normal acceleration.
- Yaw rate / roll angle.
- Pitch attitude.

(d) Load Cycles

A range-mean-pair (rainflow) counting technique which is described in detail by Ellis²⁵ is to be used. An RAE rainflow algorithm developed by Ellis was used by Holford³² for aircraft structural loads analysis. Rainflow cycle counts are to be tabulated for tail rotor torque and for all direct stress / load parameters (about 10 channels total). Cumulative fatigue life usage on critical components is to be estimated.

(c) Extreme Parameter Values

The values and times of occurrence of maximum and minimum values of stresses / loads are to be logged.

All of the above Sea King HODR program data can be presented on a flight-by-flight basis and the results integrated across flights of the same sortie type. WHL did not indicate a requirement to collate data on the basis of flight regime in its data analysis proposal³⁵. The results of the complete set of sortie spectra can be integrated to provide the fleet mission spectra.

8.9.3 Graphical Presentation and Archiving of Collated Data

Graphical presentation of the collated data can aid the appreciation of the salient features of the load spectra etc., and the extent to which the patterns stabilize over time. Parameter-time spectra can be readily presented in graphical form. A typical two-dimensional plot is shown in Fig. 3a and a three-dimensional plot in Fig. 3b.

Some means of archiving the data for future re-examination (if required), is desirable. If the data are recorded during flight in unprocessed form it would be prudent to retain data in that form as well as the collated form. In one of the data system options being considered for the S-70A-9 Black Hawk MSA program, transcription of the recorded data to optical disk media is proposed.

8.9.4 Data Analysis

The data collated as indicated in Secs. 8.9.1 and 8.9.2 are in a form which the manufacturer can use, in combination with component fatigue strength data, for estimating safe fatigue lives applicable to critical components for the operator's usage spectrum.

9. CHOICE OF DATA SYSTEM

A number of data systems which can be considered for use in helicopter fatigue substantiation programs are available commercially. Because requirements for different helicopters vary, special applications software and signal conditioning equipment would normally be required to match a general purpose data system to a specific application.

The cost of setting up a suitable data system (including airborne and ground support elements), and of using and maintaining it during the period of data collection, is the major cost element associated with a helicopter component fatigue substantiation program. The choice of a data system will depend on a number of factors such as:

- (a) Technical merit.
- (b) Ease of modifying (changing software etc.).
- (c) Cost.

- (d) Supportability (capability of available personnel to maintain it for the duration of the measurement program).
- (e) Compatibility considerations (eg. need for data collection to be compatible with data collection and collation software developed for previous programs).
- (f) Other applications for the data (eg. there may be a need to retain a version of the data in relatively unprocessed form for other purposes).

The merits of alternative data systems, based largely on commercially available elements, are being evaluated for the Australian Black Hawk MSA application.

10. CONCLUDING REMARKS

Assumed flight patterns and loading spectra usually form one of the major inputs for the manufacturer's estimation of in-service fatigue lives and designation of fatigue design lives for critical components in helicopters. Actual helicopter usage may depart significantly from the assumed usage and hence it is widely accepted that actual flight and load patterns be measured during service to allow the promulgated fatigue lives to be substantiated. A number of in-service measurement programs to substantiate helicopter component fatigue lives is being undertaken in various countries at this time. For these programs appropriate parameter measurements are made in a small number of fleet aircraft for a limited period, with 200 hours of good data being a representative requirement per instrumented aircraft while the aircraft is engaged in normal duties. Typically about 30 to 50 parameters relating to flight control inputs, aircraft responses, loads in selected critical components and other data need to be measured in flight. Airborne data systems for acquiring the appropriate data must be compact and reliable, and must place minimum demands on squadrons in respect of the operation of these systems. Since the manufacturer's involvement in the component fatigue substantiation process is invariably required, data needs to be collated in a manner compatible with the manufacturer's data format requirements. A provisional list of parameters to be measured in support of a component fatigue substantiation program for the S-70A-9 Black Hawk helicopter in service in Australia has been detailed. General requirements of a system for use in collecting and collating data required for substantiating/revising fatigue lives of selected components in Australian Service helicopters have been formulated.

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TABLE 1**Transmission Torque Limitations - Sea King**

Twin/Single Engine Operation	Condition	% Rated Engine Torque	% Rated Total Torque
Twin	Maximum Steady Indication	111	111
	Transient for not more than 5 sec	120	120
Single	Contingency Rating for Continuous Use on Failure of Opposite Engine	123	61.5
	Contingency Rating for 2.5 min on Failure of Opposite Engine	135	67.5
	Transient for not more than 5 sec	150	75

TABLE 2**Rotor Rotational Speed Limitations - Sea King**

Condition	Rotor Speed (RPM) Operational Limit	Rotor Speed (RPM) Transient Limit
Power On	95% - 105%	91% - 117%
Power Off	95% - 112.5 %	91% - 117%

TABLE 3

Helicopter Fatigue Substantiation Program Summary

Heli Type	Sponsor	No. of Heli	Program Status	In-flight Data Measured	Comments
Chinook	MOD	3	Finish 1986	Flight and Loads Data 33 ANL, 9 DIS Plessey Rec	Fatigue Subst with Support of Boeing Vertol and Lycoming (engines) Prg Acronym "HODR"
Sea King	MOD	5	Under way in 1990	Flight and Loads Data 55 ANL, 3 DIS Plessey Rec	Main Contractor WHL to do Fatigue Substantiation Prg Acronym "HODR"
Lynx	MOD	6	Planned	Flight and Loads Data Plessey Rec	Main Contractor WHL to do Fatigue Substantiation Prg Acronym "HODR"
UH-60A Black Hawk	US Army	2	On hold in 1990	Flight and Loads Data 25 ANL, 10 DIS CM Recorder	Main Contractor Sikorsky to do Fatigue Subst. Prog Acronym "SUM"
SH-60B Seahawk	US Navy	6	Under way in 1990	Flight and Loads Data 32 ANL, 6 DIS CM Recorder	Main Contractor Sikorsky to do Fatigue Subst. Prog Acronym "SUM"
OH-58C Kiowa	US Army	10	Under way in 1989	Flight Data, 13 ANL	Fatigue Subst by Bell Helicopter using Flight/Loads Transfer Function. Prog Acronym "HUM"
Super Puma	Bristow		Finish 1987	Flight Data 26 ANL, 12 DIS Penny & Giles Recorder	Fatigue Subst by Aerospatiale using Flight/Loads Transfer Function

Abbreviations:

ANL : Analogue (recorded as 10 or more bits after conversion)
 DIS : Discrete (recorded as 1 bit)
 CM : Canadian Marconi
 WHL : Westland Helicopters Limited
 HODR : Helicopter operational data recording
 HUM : Helicopter Usage Monitoring
 SUM : Structural usage monitoring

TABLE 4**Proportional Time for Various Sorties in Particular Fleet**

Type of Flying	Flight Codes	% Flying Time
Anti-Submarine Warfare	ASW	20
Anti-Submarine Exercise	CASX	15
Search and Rescue	SAR	5
Sonar Dunking	DUNK	11
Flying Practice	GFP, OFT, SCT	18
Instrument Flying Practice	IFP, IRT, NAVX	8
Test Flights	FMTF, CTF, MTF	10
Miscellaneous	Various Codes	13

TABLE 5
Sea King HODR Parameters

CATEGORY	PARAMETER	NO. CHS	SAMPLING RATE
Configuration	All-up-weight Centre of gravity position	* *	
Ambient Conditions	Pressure altitude Outside air temperature	1 1	1 Hz 1 Hz
Flight Control Inputs	Collective pitch lever position Cyclic pitch lateral position Cyclic pitch longitudinal position Yaw pedal position	1 1 1 1	8 Hz 8 Hz 8 Hz 8 Hz
Flight Control Servo Responses	Collective servo position Cyclic lateral servo position Cyclic longitudinal servo position Yaw servo position	1 1 1 1	8 Hz 8 Hz 8 Hz 8 Hz
Aircraft Responses	Pitch attitude Roll attitude Heading Aircraft normal acceleration - fore Aircraft normal acceleration - aft Aircraft fore-aft acceleration Aircraft lateral acceleration - fore Aircraft lateral acceleration - aft Forward speed	1 1 1 1 1 1 1 1 1 1	8 Hz 8 Hz 8 Hz 8 Hz 8 Hz 4 Hz 4 Hz 4 Hz 4 Hz
Engines	Gas generator speed Power turbine speed Power turbine inlet temperature Compressor delivery air temperature Compressor delivery air pressure Inlet guide vane angle Fuel flow rate Fuel temperature	2 2 2 2 2 2 2 2	2 Hz 2 Hz 2 Hz 2 Hz 2 Hz 1 Hz 1 Hz 1 Hz
Powertrain	Main rotor RPM Engine torque Tail driveshaft torque Main gearbox oil temperature	1 2 1 1	4 Hz 4 Hz 8 Hz 1 Hz
Rotor Systems	Main rotor shaft upper stress (V) Main rotor shaft lower stress (V) Main rotor top hub plate stress (M,V) Main rotor rotating star arm stress (M,V) Tail rotor spider arm (V)	1 1 2 2 1	2 Hz 2 Hz 2 Hz 2 Hz 8 Hz
Flight Control System	Tail rotor bell crank lever load (M)	1	8 Hz
Airframe	Fwd main lift frame port side stress (M,V) Aft main lift frame stbd side stress (M,V) Tail pylon hinge stress (M,V) Underslung load	2 2 2 1	8 Hz 8 Hz 8 Hz 4 Hz
Discretes	Auto flt control syst (AFCS) on/off Undercarriage scissors switch on/off Load event	1	1 Hz
Other	Elapsed time (via synch recording) Takeoff time/date Sortie type Aircraft tail number	0 * * **	 1 Hz
Notes: (M) = mean * = data logged by pilot (V) = vibratory ** = not sampled, implanted in recorded data sequence			

TABLE 6

List of US Army UH-60A Black Hawk SUM Program Inputs

CATEGORY	PARAMETER	NO. CHS	ACQ. RATE	INPUT TYPE
Ambient Conditions	Barometric altitude	1	4 Hz	DC1
	Radar altitude	1	4 Hz	DC1
	Outside air temperature	1	4 Hz	DC2
Rotor Pitch Control Inputs	Collective pitch lever position	1	16 Hz	DC1
	Cyclic pitch lateral position	1	16 Hz	DC2
	Cyclic pitch longitudinal position	1	16 Hz	DC1
	Yaw pedal position	1	16 Hz	DC2
Aircraft Responses	Pitch attitude	1	4 Hz	SYN
	Roll attitude	1	4 Hz	SYN
	Heading attitude	1	4 Hz	SYN
	Forward speed	1	4 Hz	DC1
	Barometric rate of climb	1	4 Hz	DC1
	Yaw rate	1	4 Hz	DC1
	Aircraft vertical acceleration	1	4 Hz	DC2
Engines	No. 1 engine torque	1	4 Hz	DC1
	No. 2 engine torque	1	4 Hz	DC1
Powertrain	Main rotor RPM	1	4 Hz	PR
Rotor Systems	Main rotor shaft bending stress (M,V)	2	4 Hz	DC3
	Main rotor blade droop stop load	1	4 Hz	DC3
Rotor Pitch Control System	Forward longitudinal servo load (M,V)	2	16 Hz	DC4
	Aft longitudinal servo load (M,V)	2	16 Hz	DC4
	Lateral servo load (M,V)	2	16 Hz	DC4
	Stationary scissors load (M,V)	2	16 Hz	DC4
Discretes	Weight on wheels indication	1	4 Hz	DIS
	Caution lights (9 lines)	9	2 Hz	DIS
<p>Input Types:</p> <p>SYN = 4-wire synchro 26/11.8 VRMS 400 Hz from standard aircraft system.</p> <p>DC1 = 2-wire DC signal from standard aircraft system.</p> <p>DC2 = 2-wire DC signal from specially fitted sensor (other than strain gauge).</p> <p>DC3 = 2-wire DC signal from specially conditioned signal from strain gauge specially fitted to rotating component.</p> <p>DC4 = 2-wire DC signal from specially conditioned signal from strain gauge specially fitted to non-rotating component.</p> <p>PR = 2-wire pulse rate signal 0 - 16527.9 Hz (0 - 150% range) from standard aircraft system.</p> <p>DIS = Discrete 28 VDC on/off signal or similar from standard aircraft system.</p>				
<p>Note: M = mean V = vibratory</p>				

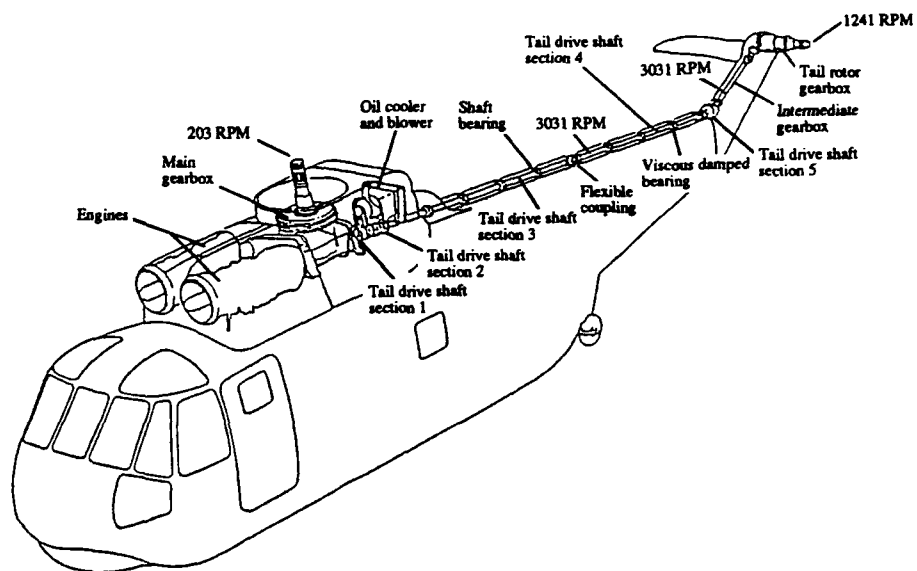
TABLE 7
List of US Navy SH-60B Seahawk SUM Program Inputs

CATEGORY	PARAMETER	NO. CHS	ACQ. RATE	INPUT TYPE
Ambient Conditions	Barometric altitude	1	4 Hz	DC1
	Radar altitude	1	4 Hz	DC1
	Outside air temperature	1	4 Hz	DC2
Rotor Pitch Control Inputs	Collective pitch lever position	1	16 Hz	DC1
	Cyclic pitch lateral position	1	16 Hz	DC2
	Cyclic pitch longitudinal position	1	16 Hz	DC1
	Yaw pedal position	1	16 Hz	DC2
Aircraft Responses	Pitch attitude	1	4 Hz	SYN
	Roll attitude	1	4 Hz	SYN
	Heading attitude	1	4 Hz	SYN
	Forward speed	1	4 Hz	DC1
	Barometric rate of climb	1	4 Hz	DC1
	Yaw rate	1	4 Hz	DC1
	Aircraft vertical acceleration	1	4 Hz	DC2
Engines	No. 1 engine torque	1	4 Hz	DC1
	No. 2 engine torque	1	4 Hz	DC1
Powertrain	Main rotor RPM	1	4 Hz	PR
Rotor System Loads	Main rotor shaft bending stress (M,V)	2	4 Hz	DC3
	Main rotor blade droop stop load	1	4 Hz	DC3
	Tail rotor stabilator control load	1	80 Hz	DC4
	Bladefold stress	1	32 Hz	DC4
Rotor Pitch Control System Loads	Forward longitudinal servo load (M,V)	2	16 Hz	DC4
	Aft longitudinal servo load (M,V)	2	16 Hz	DC4
	Lateral servo load (M,V)	2	16 Hz	DC4
	Stationary scissors load (M,V)	2	16 Hz	DC4
Undercarriage and Airframe Signals	Main landing gear oleo load (left)	1	32 Hz	DC4
	Main landing gear oleo load (right)	1	32 Hz	DC4
	Tail landing gear oleo load	1	32 Hz	DC4
	Tail pylon fold load	1	32 Hz	DC4
	RAST probe load	1	32 Hz	DC4
	Wheel spin rate	1	4 Hz	
Discretes	Weight on wheels indication	1	4 Hz	DIS
	Bladefold occurrences	1	2 Hz	DIS
	Pylon fold occurrences	1	2 Hz	DIS
	Rotor brake applications	1	2 Hz	DIS
	External lift occurrences	1	2 Hz	DIS
	RAST probe use occurrences	1	2 Hz	DIS
<p>Input Types:</p> <p>SYN = 4-wire synchro 26/11.8 VRMS 400 Hz from standard aircraft system.</p> <p>DC1 = 2-wire DC signal from standard aircraft system.</p> <p>DC2 = 2-wire DC signal from specially fitted sensor (other than strain gauge).</p> <p>DC3 = 2-wire DC signal from specially conditioned signal from strain gauge specially fitted to rotating component.</p> <p>DC4 = 2-wire DC signal from specially conditioned signal from strain gauge specially fitted to non-rotating component.</p> <p>PR = 2-wire pulse rate signal 0 - 16527.9 Hz (0 - 150% range) from standard aircraft system.</p> <p>DIS = Discrete 28 VDC on/off signal or similar from standard aircraft system.</p>				
<p>Note: M = mean V = vibratory</p>				

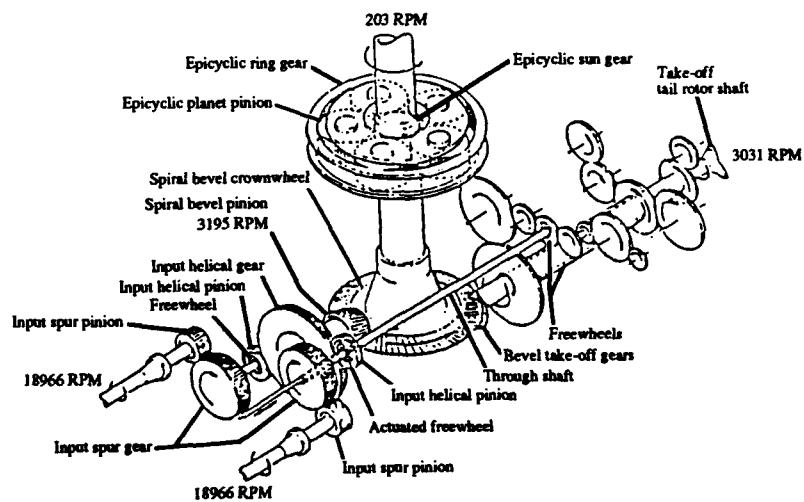
TABLE 8

Provisional List of Australian S-70A-9 Black Hawk MSA Program Inputs

CATEGORY	PARAMETER	NO. CHS	ACQ. RATE	INPUT TYPE
Ambient Conditions	Barometric altitude	1	1 Hz	DC1
	Radar altitude	1	1 Hz	DC1
	Outside air temperature	1	1 Hz	RT
Rotor Pitch Control Inputs	Collective pitch lever position	1	16 Hz	DC1
	Cyclic pitch lateral position	1	16 Hz	DC2
	Cyclic pitch longitudinal position	1	16 Hz	DC1
	Yaw pedal position	1	16 Hz	DC2
Aircraft Responses	Pitch attitude	1	4 Hz	SYN
	Roll attitude	1	4 Hz	SYN
	Heading attitude	1	4 Hz	SYN
	Forward speed	1	4 Hz	DC1
	Barometric rate of climb	1	4 Hz	DC1
	Yaw rate	1	4 Hz	DC1
	Aircraft vertical acceleration	1	4 Hz	DC2
Engines	No. 1 engine torque	1	4 Hz	DC1
	No. 2 engine torque	1	4 Hz	DC1
Powertrain	Main rotor RPM	1	4 Hz	PR
Rotor System Loads	Main rotor shaft bending stress (M,V)	2	4 Hz	DC3
	Main rotor blade droop stop load	1	4 Hz	DC3
Rotor Pitch Control System Loads	Forward longitudinal servo load (M,V)	2	16 Hz	DC4
	Aft longitudinal servo load (M,V)	2	16 Hz	DC4
	Lateral servo load (M,V)	2	16 Hz	DC4
	Stationary scissors load (M,V)	2	16 Hz	DC4
Discretes	Weight on wheels indication	1	4 Hz	DIS
	Rotor brake applications *	1	2 Hz	DIS
	External lift occurrences *	1	2 Hz	DIS
Input Types: SYN = 4-wire synchro 26/11.8 VRMS 400 Hz from standard aircraft system. DC1 = 2-wire DC signal from standard aircraft system. DC2 = 2-wire DC signal from specially fitted sensor (other than strain gauge). DC3 = 2-wire DC signal from specially conditioned signal from strain gauge specially fitted to rotating component. DC4 = 2-wire DC signal from specially conditioned signal from strain gauge specially fitted to non-rotating component. RT = 4-wire specially fitted resistance thermometer without signal conditioner. PR = 2-wire pulse rate signal 0 - 16527.9 Hz (0 - 150% range) from standard aircraft system. DIS = Discrete 28 VDC on/off signal or similar from standard aircraft system.				
Notes: * = Included in US Navy SUM, not in US Army SUM. M = mean V = vibratory				



(a) Arrangement of rotating components



(b) Layout of main rotor gearbox (MRGB)

FIG. 1. SEA KING HELICOPTER ROTATING MACHINERY

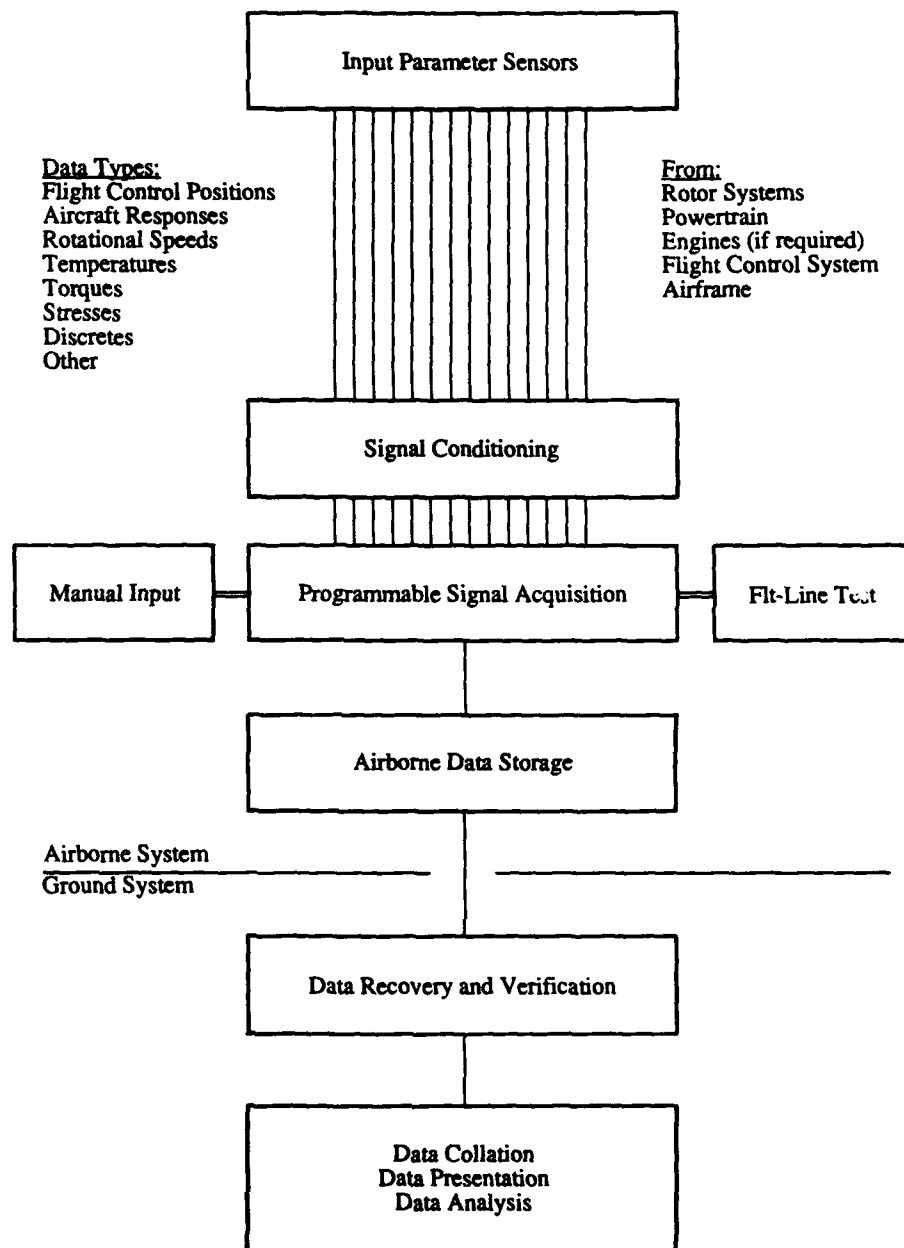
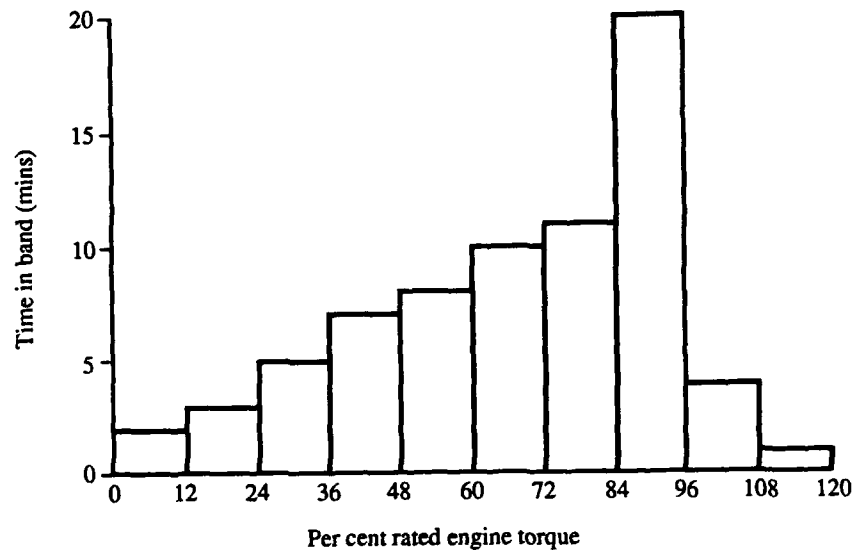


FIG. 2. DATA SYSTEM FOR MISSION SEVERITY ANALYSIS

(a) Two Dimensional Plot



(b) Three Dimensional Plot

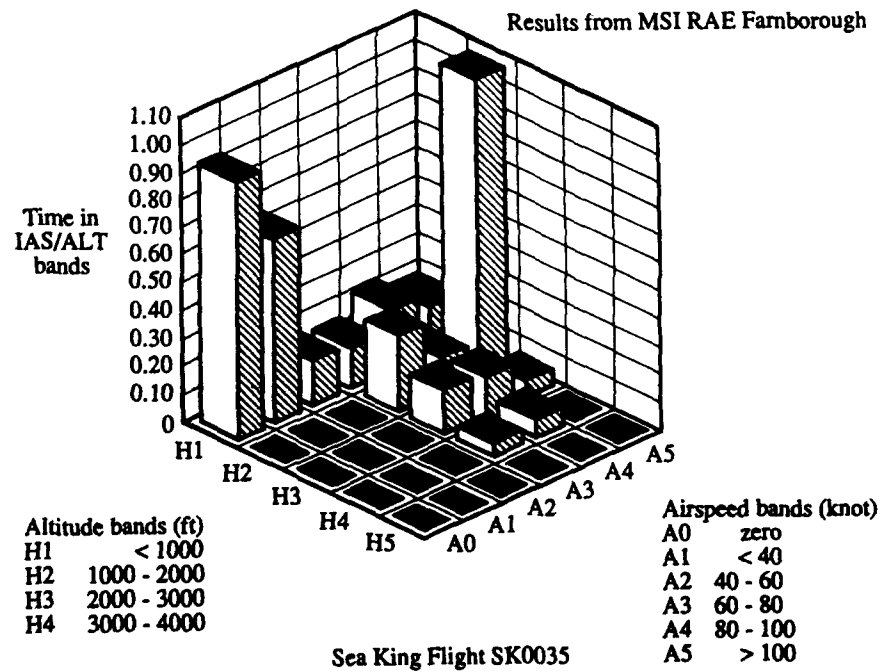


FIG. 3. PARAMETER-TIME SPECTRA

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16. ABSTRACT <i>An Australian Defence Force requirement has been defined to provide in-country capability to support component fatigue life substantiation in selected Australian fleet helicopters, with initial application to the S-70A-9 Black Hawk helicopter operated by the Australian Army. The implications of this requirement are examined, and the need to assess the severity of the spectrum of normal missions for the selected aircraft fleet is supported. Justification for a program to assess mission severity from measurements of flight regime recognition data and loads in selected components in a sample of fleet aircraft, is provided. A program to substantiate the fatigue lives of selected Black Hawk helicopter components, subject to significant in-service loads, is outlined. The general requirements of airborne and ground data systems required in support of the program are examined.</i>			

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